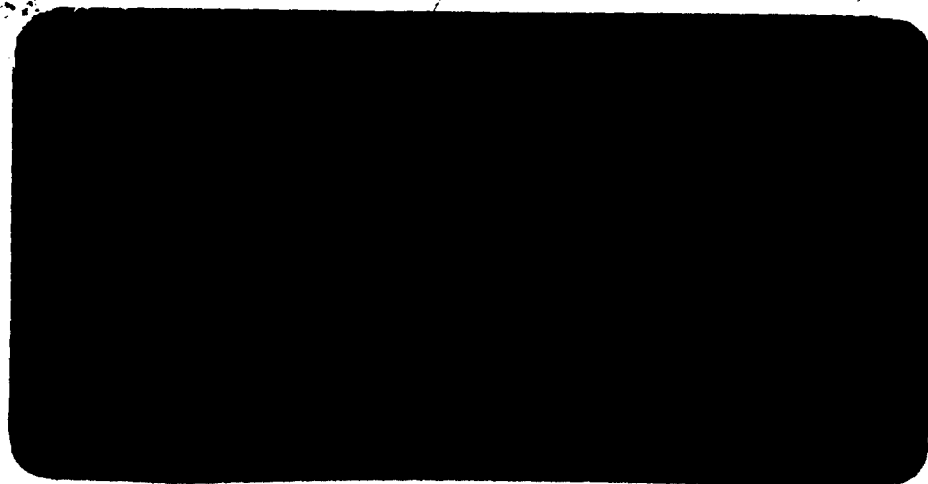


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AIRMAT MATERIALS INVESTIGATION

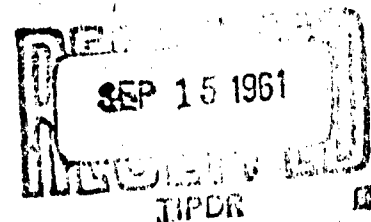
OF ONE-PLACE INFLATOPLANE

GA-468

CONTRACT Nonr 2368 (CO)

GER 10270

30 June 1961



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SUMMARY

The Goodyear Aircraft Corporation, in accordance with paragraph 7 of Amendment 8 of the Office of Naval Research Contract NOnr 2368(00), tested Airmat\* fabrics to determine the characteristics of these materials used in the construction of the GA 468 Inflatoplane\*\*. Physical properties of the Airmat cockpit, wing, and empennage surface materials are presented from rotoflex, creasing permeability, tear strength, cyclic loading and cylinder elongation tests. Additionally, wing surface materials and drop threads were subjected to quick break, dead load-time, and panel burst tests. Tests were also conducted on used aircraft to determine the effects of packaging and to establish the aircraft service life based on application of the limit load. Physical tests substantiate the ability of the material to withstand mechanical abuse; dead load and panel burst tests indicate material strength degradation with age and use.

The Inflatoplane service life, as determined by the limit load test, is conservatively calculated as being a minimum of 12,000 hours, surpassing the 7500 hour minimum required of a vehicle of this category. Mechanical abuse resulted in only minor materials degradation and does not significantly reduce the vehicles service life. Substantiating this is (1) the two to three percent increase in leak rate resulting from 75 packaging operations, (2) the minor increase in permeability of hydrogen after rotoflex and cyclic loading and (3)

-----  
\*TM Goodyear Tire and Rubber Co., Akron, Ohio

\*\*TM Goodyear Aircraft Corporation, Akron 15, Ohio

material strength after rotoflex and creasing. Permeability to hydrogen is not increased by rotoflexing; however, it does increase after application of a cyclic load of 10,000 cycles by a factor of 3 to 5 for the cockpit and empennage and a factor of 10 for the wing material, but does not increase notable thereafter up to 100,000 cycles. After rotoflexing and creasing material strength is reduced by approximately 2 to 5 percent for the cockpit and empennage and 7 percent for the wing.

Except for the cockpit material of the physical tests and the new material used in the dead load and panel burst tests, all specimens were fabricated of Airmat materials from vehicles in excess of four years old which had been subjected to loads encountered during demonstration and/or test programs conducted during this period. Hence, the data presented substantiates the ability of these fabrics to perform as a structural material after significant aging and time under load.

SECTION I - INTRODUCTION

The GA 468 Inflatoplane derives many unique capabilities from the utilization of Airmat as a major structural component. The aircraft cockpit, empennage, and wing are all constructed of Airmat providing a lightweight, high strength vehicle possessing packagability, overload recovery, flotation, ease of repair, and logistic characteristics exceeding those offered by other vehicles of this type.

Airmat consists of two layers of fabric impregnated by an elastomer or resin to withstand pressurization, joined by drop threads extending between the upper and lower fabric surfaces. When pressurized, an Airmat section attains a pre-determined shape as established by the lengths of the drop threads (see Figure I-1).

Since Airmat structures maintain their structural integrity by their ability to withstand pressurization, it is necessary that they maintain this pressure holding capability for a reasonable period of time. Detrimental to this capability are two factors (1) the natural degradation of the material with age, and (2) the mechanical abuse of the material by the otherwise advantageous feature of being inflated, deflated, packaged into a small space, and reinflated again for use.

In connection with the development of the Inflatoplane, the question arose as to how these detrimental factors influence the service life of the structure. In response to Amendment 8 of Contract NONr 2368 and in order to determine this influence, a test program for an "Airmat Materials Investigation of the GA 468 Inflatoplane" was initiated.

The test program consisted of:

- (1) panel burst tests
- (2) packaging tests
- (3) fabric life tests under the 2.5 g limit load
- (4) physical tests of surface materials to include Rotoflex, Creasing, Tear Strength, Seam Strength, Cylinder elongation and Cyclic loading tests
- (5) dead load tests on the surface fabric and drop threads of the Airmat wing materials

In conducting the tests of items 1, 4, and 5, specimens of both new and used materials were tested for comparative purposes. For evaluating items 2 and 3, airplanes were selected which had previously been subjected to a large number of structural, wind tunnel, and/or flight tests.

The purpose, method, results, and significance of each of these tests are found in the following sections of this report.

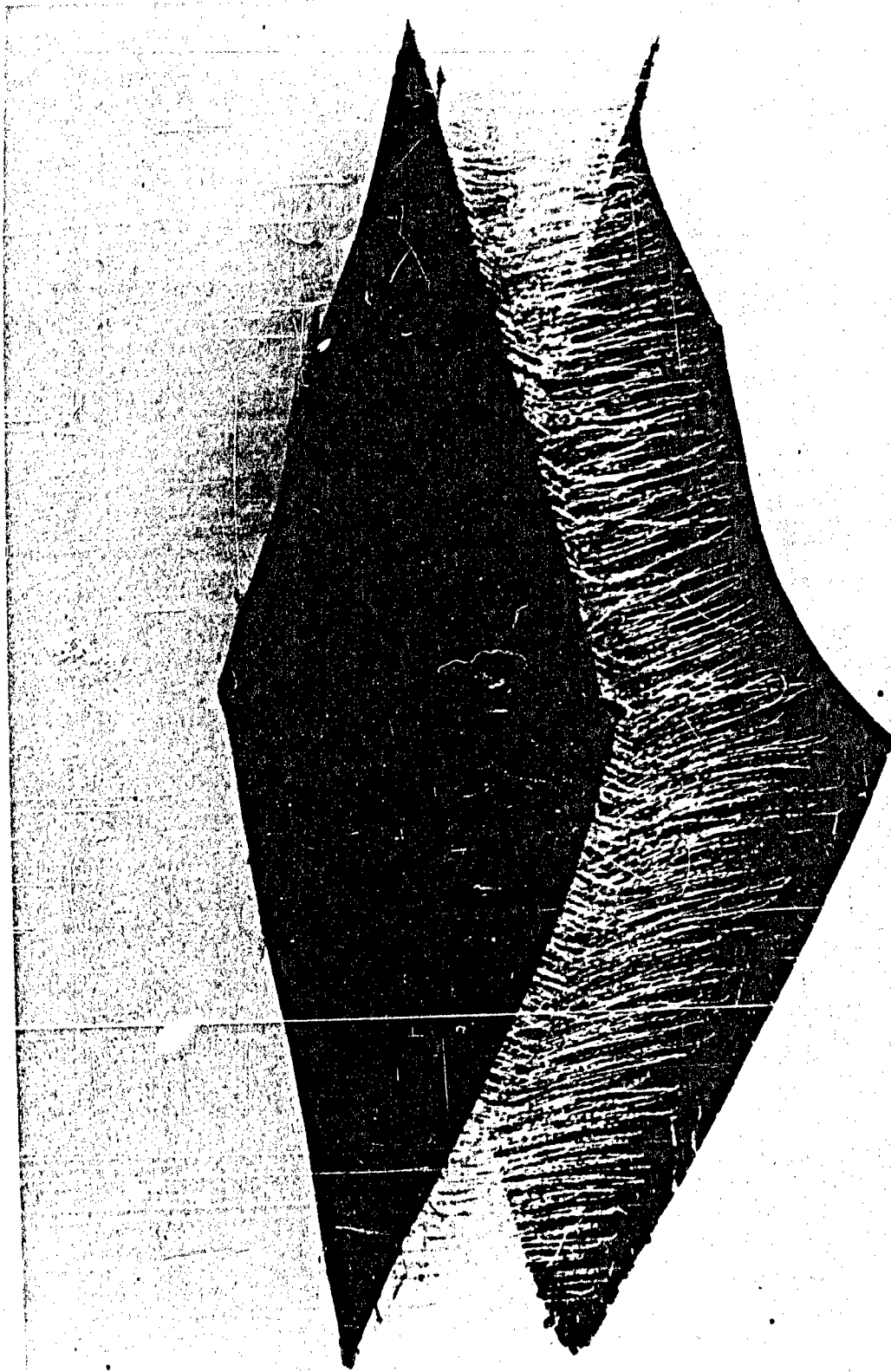


Figure I-1 Typical Airmat Cross Section

SECTION II - PANEL BURST TESTS

A. GENERAL

A pressure safety factor of 4.0 was established as the criterion of the Inflatoplane design. (Reference IV-1 Sec. 1.00.050) Based upon this factor of safety and the recommended operating pressure for the Inflatoplane of 7 psi, the Inflatoplane components should have an internal pressure ultimate load capability of 28 psi minimum.

One method of testing the pressure safety factor is by bursting representative panels. For this type of test, the panels are fabricated to simulate the structure and are inflated rapidly until a panel burst is experienced. In this manner the average burst pressure is obtained and, since pressure safety factor =  $\frac{\text{average burst pressure}}{\text{operating pressure}}$ , a check of the design pressure

safety factor is obtained. The wing of the Inflatoplane being the most critical section, it was decided that panel burst tests should be concerned primarily with panels fabricated from wing sections.

During the development phase of the Inflatoplane, a series of panel burst tests were run on 4' x 4' panels of the Airmat wing material. The average burst pressure of these panels was 30 psi and the resulting pressure safety factor,  $30/7$  or 4.28. These numbers have been presented as control values for direct comparison with the results of the repeat tests.

Two factors were present which justified repeating panel burst tests as part of this Inflatoplane Materials Investigation Study. Material strength

Degradation with aging and the presence of "hard wrinkles" in the Airmat surface plies were the two factors which could be evaluated with data obtained from panel burst tests. "Hard wrinkles" in Airmat, defined as a lapping or bunching of the Airmat surface material under the bias cover ply, occurs during the doubling process when the bias cover plies are applied. The structural effect of these wrinkles has previously been evaluated in cylinder burst and strip tensile tests as reported in Enclosure 4 of reference II-1. The presence of "hard wrinkles" led to the scrapping of four (4) Inflatoplane wing panels as reported therein. However, detection and hence elimination of all such wrinkles in new material is (1) quite difficult since they are internal and (2) unnecessary, based on the results of the previously mentioned tests.

Hence, to determine the strength reduction with age and to further justify the decisions on "hard wrinkles" listed in reference II-1, panel burst tests on Airmat wing sections were conducted.

**B. APPARATUS AND PROCEDURE**

Six Airmat panels were fabricated of the A-350 Inflatoplane wing material, three of new material, and three of material from a used wing (serial No. 4111). The material was cut such that all panels had a nine-inch drop yarn length, were basically square in planform, and exceeded two feet in both length and width. The open-ends were sealed by seam-closing. To prevent failure in the seams, particular care was taken in rounding the corners and in making the seams.

Unreinforced hard wrinkles were present in each of the panels fabricated of both new and used material. By comparing these results with previous panel burst data of new material without "hard wrinkles" a direct evaluation should be possible for determining (1) the effects of "hard wrinkles" alone and (2) the combined effects of "hard wrinkles" plus aging. It should also be noted that the used samples were made from the wing of the aircraft which had previously been used for both demonstration purposes and for the fabric life studies described in Section IV. Hence, in addition to "hard wrinkles" and aging, this wing had experienced abuse equivalent to the anticipated life of the aircraft.

Since fabric strength is a function of time under load, it was the purpose of these tests to increase the pressure rapidly to burst pressure, thereby obtaining the true burst pressure of the material.

#### C. RESULTS AND DISCUSSION

All panels were pressurized and burst. All failed in the surface plies rather than the drop threads or seams; hence, all breaks are considered good. The results are presented in Table II-1.

TABLE II-1

Airmat Panel Burst Tests

Material; A-350 Airmat Wing

Burst Pressure  
(psi)

New

1st panel	29.5
2nd panel	31.5
3rd panel	27.5
	<hr/>
	29.5 Avg.

Used

1st panel	27.5
2nd panel	23.5
3rd panel	23.0
	<hr/>
	24.7 Avg.

The effect of hard wrinkles on the new A-350 wing is considered negligible as the difference of average values ( $30 - 29.5 = 0.5$  psi) is within the scatter of the test data. Computing a factor of safety for the new material using the average burst pressure, it is  $29.5/7 = 4.22$  which is in excess of the required safety of 4.0.

In comparing the combined effects of aging and hard wrinkles on the Airmat A-350 wing material is observed that the factor of safety is reduced to  $24.7/7 = 3.53$  by using the average burst pressures. This is about an 18 percent reduction in safety factor from the control value of 4.28 and about 12 percent below the 4.0 value required. Careful inspection of the panels following the test revealed that most failures did originate in the area of a hard wrinkle.

#### D. CONCLUSIONS

Based on the results of the panel burst tests, the existence of small hard wrinkles does not significantly reduce the design pressure factor of safety of the Inflatorplane. The aircraft initially possesses a factor of safety of 4.28 which compares quite favorably with the resulting safety factor of 4.22 when hard wrinkles are present in the wing fabric. Both of these values exceed the pressure safety factor of 4.0 required of the design.

The burst tests of Airmat panels four years old which contained "hard wrinkles", and had been previously subjected to abuse equivalent to the anticipated service life of the vehicle resulted in a pressure safety factor of 3.53. Although this value is 12 percent lower than the design safety factor of 4.0, it is remarkably high considering the punishment endured prior to conducting these tests.

Although most breaks did originate with a "hard wrinkle" the data substantiates that the presence of wrinkles alone does not significantly effect the panel strength. Hence, the recommended fix of applying an additional surface ply over a wrinkled area appears most satisfactory. (reference II-1).

SECTION III - PACKAGING TESTS

A. GENERAL

An Inflatoplane derives unique capability by possessing the advantageous features of being deflatable, packagable into a small space for storage and handling, and quickly reinflatable again for use. The mechanical abuse of the materials resulting from such action has never been fully explored. In order to determine the extent of these packaging effects on the fabric materials, especially Airmat, GAC performed packaging tests as part of the Inflatoplane Materials evaluation program.

Damage to the Inflatoplane materials during packaging will result primarily from (1) abrasions or scratches inflicted by rigid hardware and components or (2) from tight creasing of the material at the folds. It is possible to establish the extent of such damage by subjecting the structure to a number of packaging and unpacking sequences while systematically recording the leak rate and computing any increase.

B. APPARATUS AND PROCEDURE

By mutual agreement between ONR and GAC a rejected Inflatoplane wing with its associated hardware was accepted as a representative substitute for a complete airplane. The only available wing was from aircraft serial number 4107 which had previously been subjected to the NASA Langley wind tunnel tests. Since the wind tunnel test program included testing to the ultimate load, this wing had experienced damage from a test failure, leaving the

basic wing material in a questionable state. Due to the condition of the material, it was uncertain at the onset whether the structure would survive many packaging operations without experiencing a considerable change in leak rate.

Following the repair of two large tears and ten minor leaks, the wing was inflated to operating pressure (approximately 14 inches of mercury) and coated with a soap solution to determine the initial state. Although leaks were found and recorded in both the upper and lower surfaces, all were minor in nature and the wing held pressure reasonably well, losing only 0.12 inches of mercury pressure in twenty minutes (see Table III-1).

The wing was then subjected to packaging and unpackaging operations in accordance with the procedure prescribed in the Inflatoplane handbook, reference III-1. The wing was folded chordwise - trailing edge to leading edge - and then folded from the tip to mid-span by wrapping the fabric around the wing-tip skid. Tight folds were made and when each semi-span was packaged at the center it was subjected to a man sitting and bouncing on the package for approximately 1 minute. The wing was then unfolded. After completing this operation five times, the wing was inflated to 14 in. of mercury and held for 15 minutes.

Preliminary investigation shows that in order to obtain consistent data this soaking period was very critical because of the creep properties of Nylon. Therefore, each time the leak rate was checked, the soaking time at the inflation pressure had to be identical. A 15 minute period was

arbitrarily chosen. After the wing was subjected to 14 in. of mercury for 15 minutes the leak rate was checked by manometer every five minutes for an additional period of 20 minutes.

The wing was then evacuated of air by a vacuum pump and subjected to five more packaging cycles after which the leak rate was again checked. This procedure was followed for a total of 15 cycles resulting in 75 separate packaging operations.

At the conclusion of the tests the wing was again coated with a soap solution to determine its condition. Careful inspection revealed one additional leak on the top surface and one on the bottom surface. The leak rate was then checked after the wing was inflated for a period of approximately 1 hour, corresponding to the soak period the wing was subjected to at the start of the tests.

### C. RESULTS AND DISCUSSION

The results of the packaging tests are recorded in Table III-1. Wing pressures are given in inches of mercury for each successive series of 5 packaging cycles through a total of 75 packagings. In each case, the zero time recording corresponds to the initial pressure after a 15 minute soak period at the aircraft operating pressure (14 inches of Mercury). Successive pressure readings correspond to an additional 5 minute time interval through a total of 20 minutes. The initial and final pressure recordings include a one-hour soak period.

An open tube manometer was used to determine the pressures; hence, reading accuracy was limited to one tenth of an inch. The second decimal place

appearing in the data table was obtained by visual extrapolation of the manometer scale. Also, since pressure differential was being obtained, slight deviations in atmospheric pressure would affect the readings. Temperature fluctuation would also effect the pressures somewhat. No attempt was made to correct the data for these three variables since only large pressure losses were of interest to the test results. Slight discontinuties which appear in the data table can be attributed to the presence of these variables.

TABLE III - 1

LEAK RATE MEASUREMENTS

		Pressure (in Hg)				
	Soaking Time	0	5 min	10 min	15 min	20 min
Initial check	1 Hr	14.00*	13.98	13.96	13.92	13.88*
After 5 folds	15 min	14.00	13.88	13.75	13.65	13.50
After 10 folds	15 min	14.00	13.90	13.70	13.55	13.40
After 15 folds	15 min	14.00	13.90	13.80	13.68	13.42
After 20 folds	15 min	14.00	13.90	13.80	13.65	13.50
After 25 folds	15 min	14.00	13.84	13.72	13.70	13.65
After 30 folds	15 min	14.00	13.90	13.75	13.58	13.45
After 35 folds	15 min	14.00	13.88	13.73	13.55	13.40
After 40 folds	15 min	14.00	13.90	13.78	13.65	13.48
After 45 folds	15 min	14.00	13.90	13.75	13.63	13.45
After 50 folds	15 min	14.00	13.85	13.72	13.62	13.45
After 55 folds	15 min	14.00	13.90	13.80	13.65	13.48
After 60 folds	15 min	14.00	13.88	13.80	13.70	13.60
After 65 folds	15 min	14.00	13.88	13.75	13.65	13.50
After 70 folds	15 min	14.00	13.85	13.65	13.55	13.42
After 75 folds	15 min	14.00	13.80	13.70	13.55	13.45
Final check	1 hr	14.00*	13.90	13.75	13.65	13.50*

\*See computations of packaging results in Part D.

#### D. CONCLUSIONS

The results indicate that after the first 5 packagings, the leak rate remained constant for all practical purposes. It can be assumed, therefore, that the two additional leaks found at the conclusion of the tests occurred during the first five folding operations. Since the diffusion rate of the wing did not deteriorate with subsequent packaging cycles and considering the initial condition of the wing, the primary cause of these leaks was probably not due to packaging but was the result of defective material. No damage to seams or reinforcement patches was evident at the end of the tests.

Comparing the pressure loss of the initial and final readings (starred table values) the results of 75 packagings are as follows:

$$P \text{ initial} = 14.00 - 13.88 = 0.12 \text{ in. mercury}$$

$$P \text{ final} = 14.00 - 13.50 = 0.50 \text{ in. mercury}$$

$$\text{increase in } \Delta P = 0.38 \text{ in. mercury}$$

or in terms of percent

$$P = (100) \frac{.38}{13.88} = 2.75\% \text{ increase in rate}$$

of pressure drop for a time interval of 20 minutes.

In light of the above test results, it is apparent that repeated packaging has little or no effect on the life of the Inflatorplane.

SECTION IV - FABRIC LIFE UNDER LIMIT LOAD

## A. GENERAL

As part of the fabric evaluation program, tests were conducted to provide data applicable for predicting an aircraft service life. To accomplish this within the scope of the minimum effort program, an endurance load test was devised whereby the limit flight load would be continuously applied to the aircraft for a period of 100 hours. Although this type of test is not in accordance with reference IV-1 (Mil-A-8866(ASG)) used for substantiating aircraft service life, fabric fatigue life is more dependent upon time-load effect rather than a cyclic load condition as is the case with metals. Therefore, by applying the design limit load for an extended period of time, it can readily be established that the aircraft develops a significant factor of safety at this most critical loading condition.

After successfully completing 100 hours of continuous limit load testing it was decided to continue the tests in an effort to provide more effective data. Testing was finally terminated after completing 336 hours without failure.

## B. APPARATUS AND PROCEDURE

The tests were conducted on a model GA 468 Inflatoplane which consisted of a wing from aircraft serial number 4111 and a fuselage, cockpit, and empennage from serial number 4108. Both of these planes had experienced considerable service prior to these life tests, having been used during the development phases for flight demonstration wind tunnel and static testing purposes.

Since these tests were to be conducted without engine, engine mount, fuel or pilot, the plane was initially weighed and the c.g. position determined.

The airplane was next inverted and suspended from the roof structure by means of a sling seven inches wide positioned around the fuselage. The suspension band was placed six inches forward of the required c.g. position due to interference with the aircraft aft wing brace cable system. To compensate for the resulting tail heaviness a 100 pound weight was added at the cockpit bulkhead station.

To simulate the 2.5 g limit load condition the following computations were made:

Inflatoplane gross weight	550 lbs
550 lbs x 2.5 g	= 1375 lbs
Wing weight = 50 lbs	
Wing inertia = 50 lbs x 2.5g =	<u>125 lbs</u>
Required wing loading	1250 lbs

The cockpit load which gives the equivalent moment to the limit load moment is 685# (Ref. IV-2, GER 9861, pg 2.06.030) i.e. 11,125 in-lbs/16.5 in. = 685# which includes a pilot weight of 240 pounds and the corresponding cockpit and instrumentation weight. Also, the limit load condition occurs when the airplane angle of attack ( $\alpha$ ) is approximately 13.8 degrees, as is shown in Ref. IV-2, GER 9861, pg 2.00.030. Figure IV-1 illustrates the loading arrangement schematically and Figure IV-2 is a three quarter photographic view of the set up. As can be seen in Figure IV-1 the airplane was suspended to simulate the proper attack angle and the desired cockpit load was

obtained by resting the nose of the plane on a pedestal supported at the proper cockpit position by a floor scale.

To simulate the 2.5 g load condition, 1250 pounds was added along the wing span at the quarter chord by applying 25 pound shot bags symmetrically at each of the following stations (measured in inches spanwise from the aircraft longitudinal centerline): 2, 5, 9, 14, 18, 22, 26, 31, 35, 39, 44, 48, 53, 58, 63, 67, 72, 77, 82, 93, 99, 105, 112, and 121.

Tape measures were fastened to the wing leading edge at six stations, i.e. approximately 14.5, 77 and 127.5 inches, symmetrically from the centerline, for wing deflection measurements.

The Inflatoplane was inflated and thereafter held at a constant pressure of 7 psi by means of a pressure regulator. Air supply was from the factory air pressure system.

To indicate the time, if failure should occur during the night when the arrangement was unattended, an electric clock was set up with two switches in its circuit. The switches were located at each wing tip and mechanically connected to the wing in such a manner that the circuit would be broken and the clock stopped in the event of a structural failure.

The test was started on 5 December 1960 when the wing load was applied after zero deflection readings were taken. Immediately after application of the load, wing deflections were again recorded. Thereafter wing deflections were recorded twice daily.

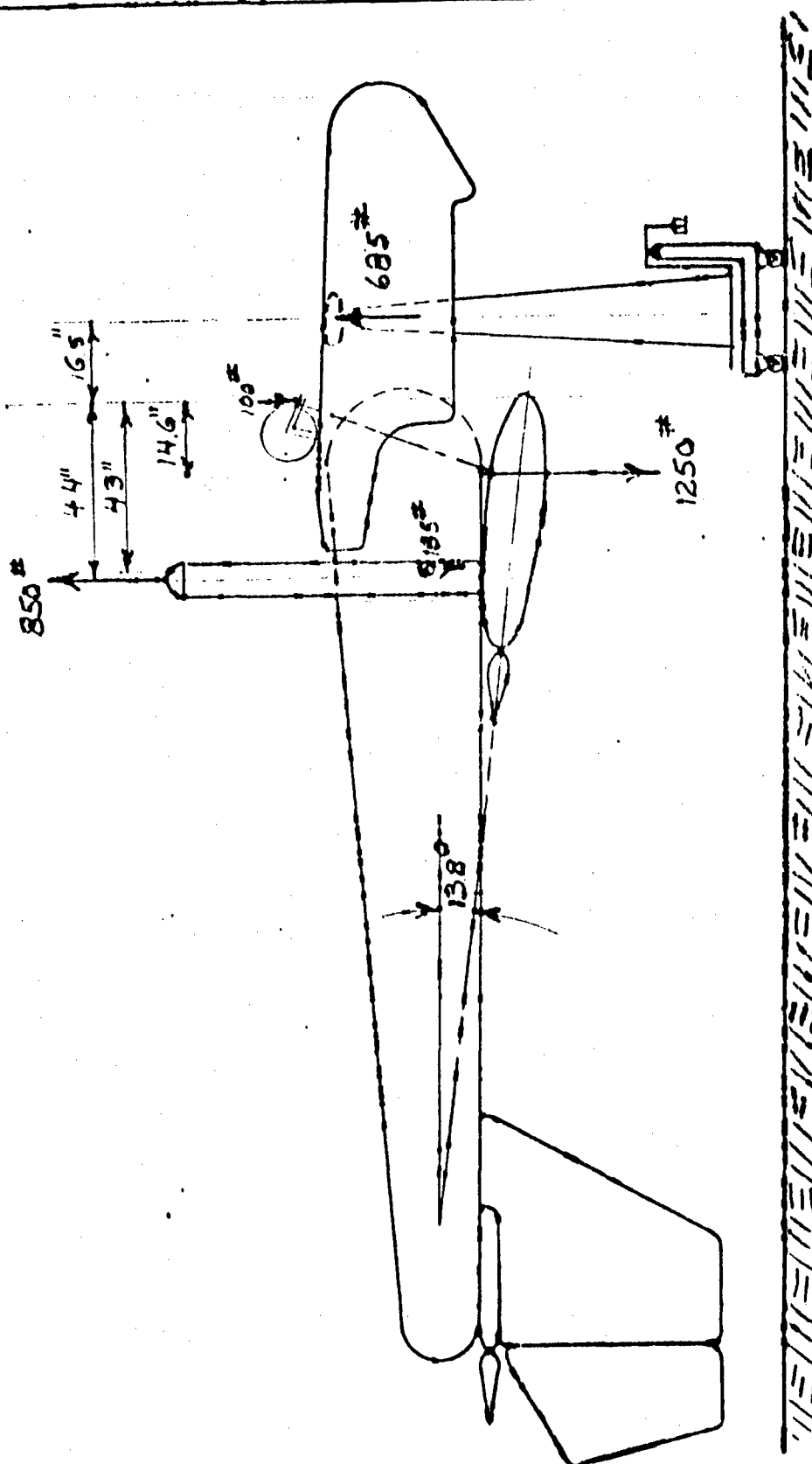


FIGURE IV-1 SCHEMATIC - LOADING ARRANGEMENT

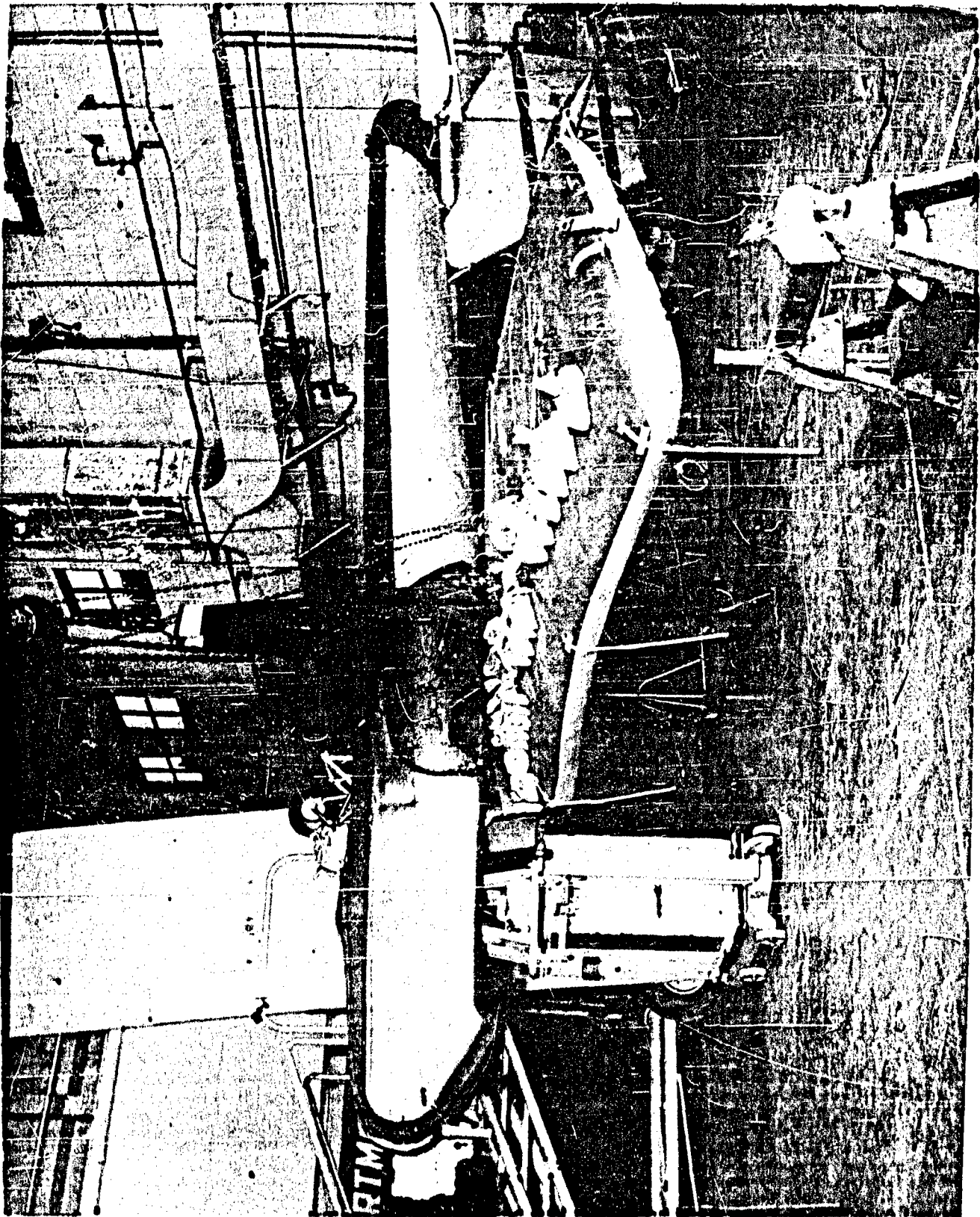


Figure IV-2 - Test Arrangement

The load was removed on 19 December 1960 - after a total of 336 hours. Deflection measurements were recorded after removing half the load and again after all load had been removed. Finally, two more readings were taken at 4 hours and 23 hours after load removal before the test arrangement was disassembled on 20 December 1960.

## C. RESULTS AND DISCUSSION

### 1. Deflections

The deflections measured during the test are presented in Table IV-1 and are shown graphically in Figures IV-3 and IV-4. Figure IV-3 shows the maximum deflection over the wing span and Figure IV-4 shows deflection versus time for the wing tips, Stations 1 and 6, and the outer brace cable points, Stations 2 and 5. In the plots of Figure IV-3 and IV-4 average values between left and right side are shown; however, as can be seen from the data of Table IV-1, the left wing showed greater deflections than the right wing. This may be attributed to either of the following reasons:

- (a) The wing of the airplane used for the test had been previously subjected to a great number of tests, which may have resulted in a permanent set particularly on the outboard left wing. Figure IV-5, which gives a front view of the test set-up before loading illustrates the presence of some permanent wing warpage.

- (b) An effort was made during the test set-up to position the wing as nearly horizontal as possible; however, the fuselage may have slipped slightly in the suspension band during loading.

## 2. Service Life Computation

An aircraft service life is normally specified by the military procuring agency. Then, in accordance with Reference IV-1, the flight maneuver spectrum is determined which indicates the frequency and intensity of the loads which are anticipated during the life of the vehicle. The responsibility of the contractor encompasses a fatigue type test arrangement whereby these anticipated loads are applied to the structure to verify its capability for withstanding the specified loads for a test period equivalent to the vehicles required service life.

Although such a program is not without merit, in application it becomes a time consuming and expensive operation above and beyond the scope of the present Inflatoplane program. Also, the fact that MIL SPEC A-8866(ASG) was written to encompass rigid structures rather than inflated fabric structures may tend to influence the validity of data obtained from conducting such a program. However, in order that some level of confidence may be established that the Inflatoplane does possess a significant service life, GAC devised an endurance test which imposed application of the flight limit load to the structure for a period of 336 consecutive hours.

In accordance with Table 1 of Reference IV-1 the nature of the Inflatoplane mission is such as to place the aircraft in category C of the flight

TABLE IV-1

Wing Deflections (inches)

Date 1960	Time Hr. Min.		Station No. Distance From Load	1	2	3	4	5	6
				127.5"	77"	14.5"	14.5"	77"	127.5"
12-5	0	10	1250#	15.48	10.65	6.02	5.23	7.13	10.24
"	1	00	"	16.39	10.95	6.12	5.38	7.44	10.89
"	5	30	"	17.03	11.27	6.34	5.58	7.93	11.99
12-6	22	30	"	17.73	11.80	6.60	5.85	8.92	12.92
"	29	30	"	17.83	11.80	6.60	5.78	8.56	12.54
12-7	46	30	"	17.82	11.80	6.59	5.80	8.56	12.58
"	53	30	"	17.88	11.85	6.60	5.80	8.60	12.69
12-8	70	30	"	17.86	11.82	6.62	5.80	8.55	12.62
"	77	30	"	18.03	11.90	6.62	5.80	8.61	12.77
12-9	94	15	"	18.08	11.96	6.68	5.88	8.68	12.85
"	101	40	"	18.30	12.07	6.71	5.88	8.80	13.06
12-10	119	30	"	18.30	12.04	6.70	5.90	8.76	13.05
"	125	35	"	18.41	12.06	6.62	5.90	8.81	13.19
"	128	00	"	18.42	12.06	6.61	5.88	8.81	13.22
12-11	141	45	"	18.40	12.10	6.68	5.88	8.80	13.18
"	150	25	"	18.42	12.08	6.75	5.90	8.80	13.13
12-12	167	00	"	18.30	12.08	6.75	5.93	8.76	13.06
"	173	40	"	18.34	12.08	6.78	5.95	8.80	13.08
12-13	190	30	"	18.38	12.11	6.79	5.98	8.84	13.12
12-14	214	40	"	18.38	12.12	6.79	5.98	8.86	13.17
12-15	238	50	"	18.43	12.13	6.79	5.97	8.86	13.21
12-16	269	30	"	18.48	12.16	6.78	5.96	8.91	13.36
12-17	291	10	"	18.53	12.14	6.78	5.98	8.88	13.30
12-18	316	45	"	18.53	12.15	6.78	5.98	8.90	11.35
12-19	336	20	1250	18.38	12.12	6.79	5.97	8.86	13.26
12-19	336	40	650#	15.45	10.12	5.96	5.16	6.80	10.11
12-19	337	00	Zero	9.68	6.71	4.19	3.38	3.28	3.01
12-19	337	25	"	8.86	6.49	4.02	3.36	3.20	3.79
12-19	341	50	"	8.83	6.49	3.84	3.05	2.70	3.09
12-20	360	45	"	7.38	5.44	3.52	2.80	2.43	2.59

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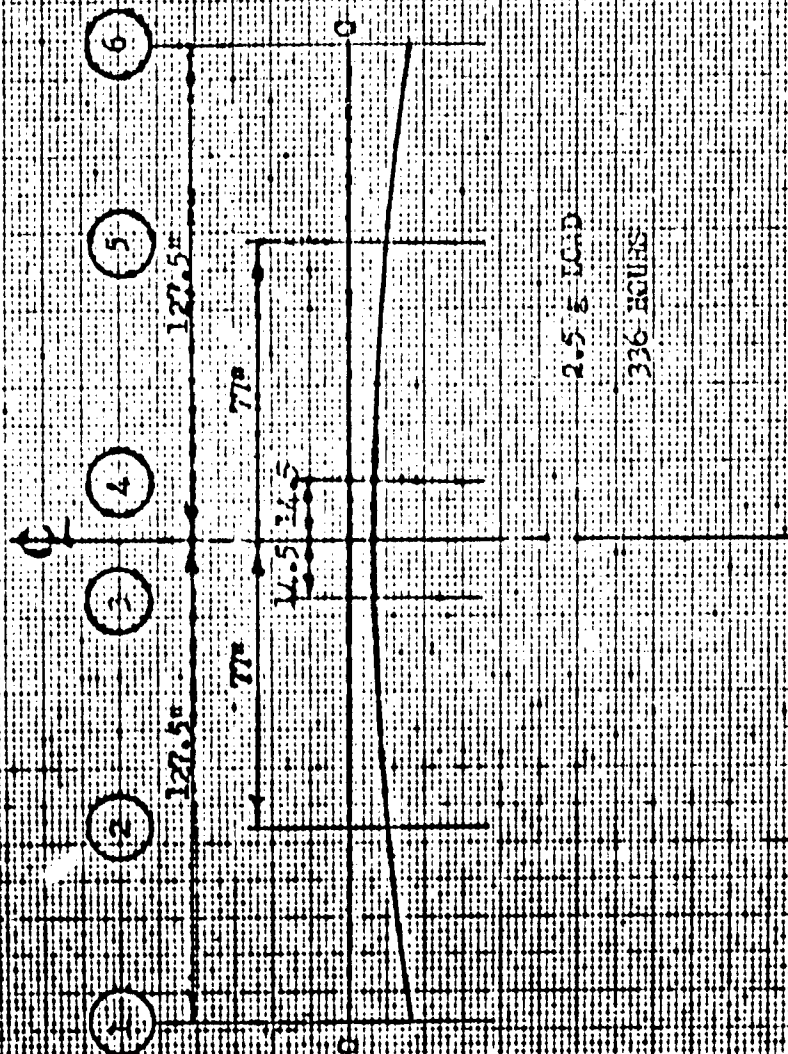


FIGURE 14-3 MAXIMUM BENDING DEFLECTION PROFILE



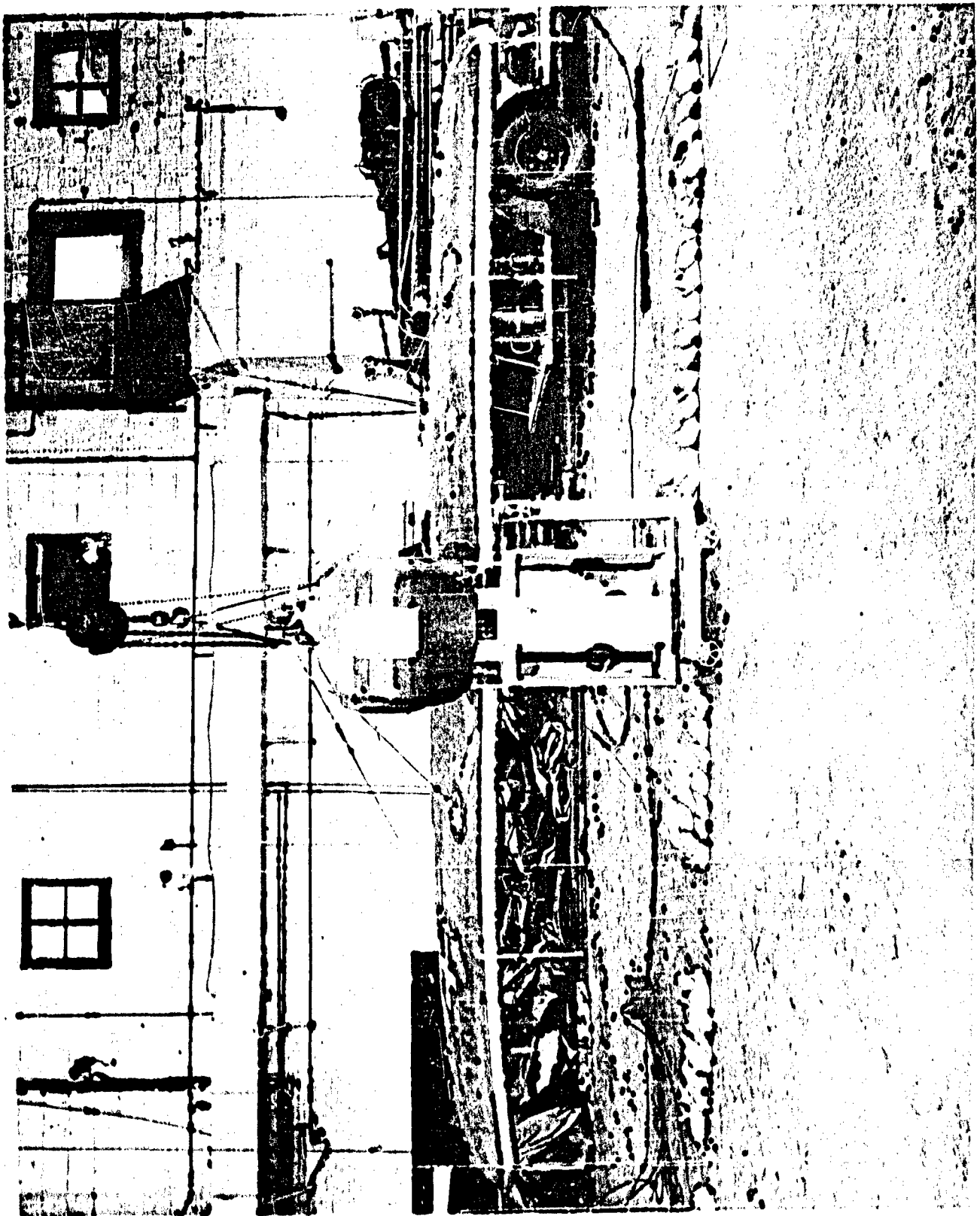


Figure IV-5 - Front View of Aircraft Before Loading

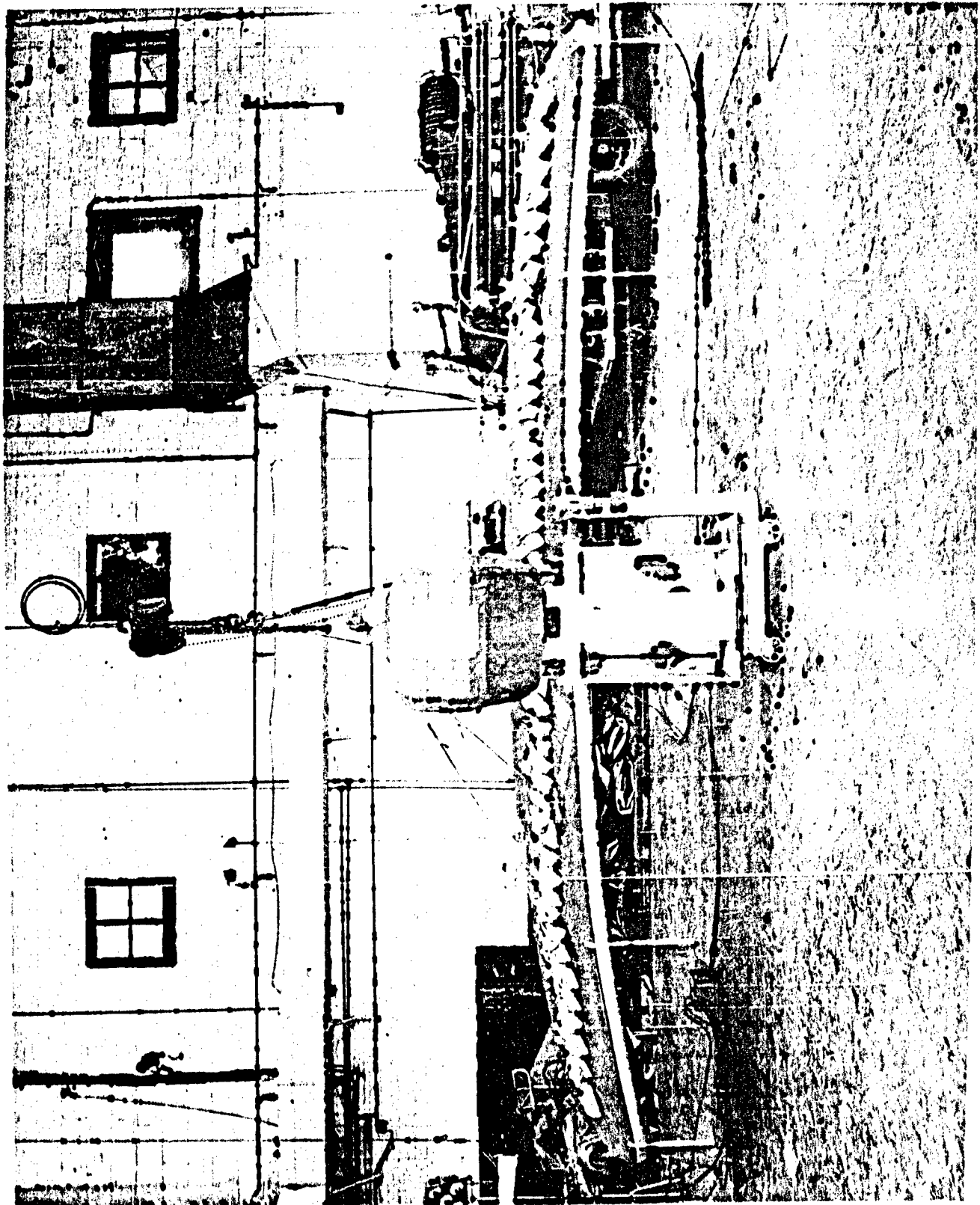


Figure IV-6 - Front View of Aircraft After Loading

maneuver spectrum which specifies a minimum service life of 7,500 hours. From Table II of Reference 1 an aircraft in Category C can expect the frequency of maneuver loads reproduced in Table IV-2 during each 1000 hours of flight time.

**TABLE IV-2**  
**Inflatoplane Flight Load Spectrum**

(1) Percent of Limit Load	(2) Frequency/1000 hrs	(3) (1) x (2)
.45	10,000	4500
.55	3,000	1650
.65	1,000	650
.75	300	187.5
.85	100	85
.95	30	28.5
1.00	20	20
Totals	14,400	7121

By multiplying the percent of limit load (Col. 1) times the frequency of which this load will be applied in 1000 hours (Col. 2) it is possible to arrive at a weighted load cycle per 1000 hours of flight time (Col. 3). Hence, the percent of load cycle for application of the limit load is:

$$\frac{\text{Percent of Limit Load}}{1000 \text{ hrs}} = (100) \frac{20}{7121} = .281\%$$

or, for each 1000 hours of flight time it may be assumed that the limit load is applied for a period of 2.81 hours. Then, since the test represented 336 hours at the limit load condition it may be said that in terms of service life the Inflatoplane has the following capability

$$\frac{2.81}{1000} = \frac{336}{\text{Service Life}}$$

$$\text{Service life} = \frac{336(1000)}{2.81} = 120,000 \text{ hrs.}$$

To account for any scatter in data, a scatter factor of 10 has been arbitrarily selected. Hence:

$$\text{SERVICE LIFE} = 12,000 \text{ hours}$$

which is in excess of the 7,500 hours required of a vehicle in this category.

As an indication of the conservatism used in the above analysis it may be pointed out that although the analysis assumes that the limit load would be applied for a period of 2.81 hours for each 1000 hours of flight, Reference IV-3, "Demonstration Progress and Instrumentation Report on the GA 468 Inflatoplane," reports on actual time requirement of 1.5 sec. to perform a pullout maneuver, the limit load. Since this load is more realistically  $1.5 \times 20 = 30$  second per 1000 hours rather than the 2.81 hours used in the calculations. Hence, an additional factor of safety of  $\frac{2.81 \times 3600}{30} = 338$  has been included to compensate for the fact that a static test rather than a fatigue test was used to determine the vehicle's service life.

Also, it should be noted that these figures do not represent the limit of the capability of the aircraft since no failure occurred. Testing was finally terminated without experiencing a failure in spite of the fact that the aircraft has seen considerable use prior to initiation of the life endurance test.

#### D. CONCLUSIONS

The deflection measurements show an initial displacement of approximately 7.5 inches at the wing tips and 3.5 inches at the outer brace cable attachment point immediately upon application of the 2.5 g limit load. After initial displacement, a more gradual displacement occurs for a period of about 35 or 40 hours. Following this phase only slight fabric creep was experienced throughout the remainder of the test until the load was removed after 336 hours. Upon removal of the load, deflections again decreased quite rapidly at first and more slowly thereafter (see Figure 6) leaving a remainder of approximately 20 percent deflection 24 hours after load removal. Maximum deflection of the wing tip was 7.5 percent of the wing half span.

Aircraft service life based on the limit dead load test is conservatively calculated as 12,000 hours minimum. This figure exceeds the specified service life of 7,500 hours required of a vehicle of this classification.

## SECTION V - PHYSICAL TESTS OF SURFACE MATERIALS

### A. GENERAL

To determine the resistance of Airmat surface materials to mechanical abuse, the Inflatoplane empennage, wing, and cockpit materials were subjected to a series of tests designated as standard methods for determining losses in physical properties of fabrics. The useful life of the Inflatoplane would be seriously impaired should these materials exhibit excessive losses resulting from such tests. Aged fabric samples were selected to further stress the materials physical capabilities for withstanding such treatment.

The empennage, Code A-349, and the cockpit, Code A-351, cover fabrics are essentially the same material, a two ply straight construction varying only in depth of the Airmat section. Test values obtained from these materials are, therefore, directly comparable. The wing material, Code A-350, differs from the former two as it consists of three plies per surface; a left bias, right bias, and a straight ply. The processed fabric specifications are given in Table V-1 as extracted from enclosure 1 of Reference II-1.

Test specimens of the empennage and wing materials were from used panel sections while the specimen of the cockpit material was from aged but unused material.

## B. APPARATUS AND PROCEDURE

The materials testing facilities of the Research Laboratory of the Goodyear Tire and Rubber Company were used for performing the tests reported herein.

All tests were performed in accordance with the procedures as outlined in Reference V-1 (MIL-C-21189(AER)). This military specification was written and approved by the Bureau of Aeronautics, Department of the Navy, as a standard for evaluating ZPG-2 and ZPG-2W laminated airship envelope cloth; hence, the individual test procedures will not be repeated here. The reader is referred to the above mentioned military specification for additional information pertaining to the test apparatus and procedures used in obtaining these results.

## C. RESULTS AND DISCUSSION

The physical test data obtained as a result of these tests are given in Table V-2. Since only a limited amount of fabric was available for these tests, the strip tensile method was selected for obtaining the material breaking strength and ultimate elongation. Although this method is approved for fabric without bias plies, lower strength values usually are obtained as compared to the cylinder burst method. This is caused by slight misalignment of the yarns and the inability of the bias plies to carry their share of the load when conducting strip tensile tests, conditions which do not influence cylinder burst strength values. Hence, as would be expected, material breaking strengths for the older materials tested here

are not as high as those reported in Table V-1 (obtained from cylinder burst tests of new material) but are suitable for comparative purposes when evaluating the degradation of physical properties resulting from mechanical abuse.

TABLE V-1

Airmat Fabric Specifications Inflatorplane

		Wing	Cockpit	Empennage Aileron & Flap
1.	Classification			
2.	Goodyear Code	A350	A351	A349
3.	Outside Color	Plain	Plain	Plain
4.	Number of Plies	3	2	2
5.	Construction (outside to inside)	(1)	(1)	(1)
	a.) Spread (oz/sq/yd)	1.25	-----	-----
	b.) Cloth (oz/sq/yd)	1.40 BL	-----	-----
	c.) Spread	2.50	1.25	1.25
	d.) Cloth	1.40 BR	2.05 S	2.05 S
	e.) Spread	3.00	5.50	5.00
	f.) Airmat Cloth	15.00 S	8.60 S	9.25 S
6.	Nominal Weight - oz/sq yd	34.00	26.20	26.20
7.	Weight Tolerance - oz/sq yd	1.70	1.25	1.50
8.	Tensile-Min-lbs/inch-Warp	180	150	140
9.	Min-lbs/inch-Fill	174	150	140
10.	min-lbs/inch <sup>2</sup> -Pile	28	28	28
11.	Tensile Test Method (Warp & Fill)	Cyl.Burst(2)	Cyl.Burst(2)	Cyl.Burst(2)
12.	Material	Nylon	Nylon	Nylon
13.	Cloth - Outside to Inside	3523N 3523N (1)	3511N 3514N (1)	3511N 8937 (1)

(1) For Airmat Construction each side is symmetrical.

(2) For Airmat, cylinders are made from each surface with pile yarn cut away for testing.

BL - Bias Left

BR - Bias Right

S - Straight Fly

TABLE V-2

Physical Test Data on Empennage, Wing  
and Cockpit Surface Materials

Section Code No.	Empennage, Aged Used A-349		Wing, Aged Used A-350		Cockpit, Aged Unused A-351	
	<u>Warp</u>	<u>Fill</u>	<u>Warp</u>	<u>Fill</u>	<u>Warp</u>	<u>Fill</u>
1) Breaking Strength (lbs/2 inch)	486 420 475 477 492	360 366 375 383 370	340 383 377 392 412	325 342 352 343 335	445 469 480 472 450	265 286 295 299 280
Average (lbs/in)	253	185	190	169	231	142
2) Ultimate Elongation Avg. (percent)	38.9	41.2	41.7	39.2	31.4	27.0
3) Breaking Strength After Rotoflexing (lbs/2 inch)	436 469 474	392 360 346	379 398 380	338 360 301	389 494 504	314 338 310
Average (lbs/in)	230	183	193	166	231	160
4) Ultimate Elongation After Rotoflexing Average (percent)	35.0	37.8	41.9	41.0	30.3	28.8
5) Breaking Strength After Creasing	505 410 475	370 360 385	375 373 310	319 319 327	470 472 365	301 300 311
Average (lbs/in)	231	186	176	161	218	152
6) Tear Strength (lbs)	56 54 53	49 50 53	140 115 120	100 103 103	50 52 50	45 50 45
Average	54	51	125	102	51	47

TABLE V-2 (cont'd)

Section Code No.	Empennage, Aged Used A-349		Wing, Aged Used A-350		Cockpit, Aged Unused A-351	
	Warp	Fill	Warp	Fill	Warp	Fill
7) Permeability to H <sub>2</sub> (liters/m <sup>2</sup> / 24 hrs)	(1) 0.5-0.5 (2) 0.5-0.5 (3) 0.0-0.0		0.5-0.4 0.2-0.4 0.4-0.5		0.5-0.7 0.8-0.8 0.1-0.1	
8) Permeability to H <sub>2</sub> after Roto- flexing	(1) 0.5-0.5 (2) 0.2-0.3 (3) 0.2-0.4		0.1-0.2 0.2-0.5 0.5-0.5		0.6-0.8 0.5-0.7 0.3-0.5	
After 10,000 cycles						
9) Permeability to H <sub>2</sub> after Cyclic Loads	(1) 2.6-2.5 (2) 1.3-1.3 (3) 2.0-2.1		1.0-1.0 1.0-1.0 1.1-1.0		3.1-3.1 1.9-1.9 1.7-1.7	
After 100,000 cycles						
	(1) 2.1-2.1 (2) 1.7-1.8 (3) 2.2-2.8		1.1-1.0 1.1-1.1 Sample 3 damaged Testing Stopped After 10,000 Cycles.		3.1-2.8 2.1-2.1 2.9-3.0	
10) Seam Strength (lbs/2 inches)	Failure in					
	150 splicing fabric		330 seam		265 seam	
	175 " "		328 seam		283 seam	
	155 " "		360 seam		262 seam	
	183 " "		362 seam		300 seam	
	125 " "		330 fabric		285 seam	
	174 " "		317 seam		299 seam	
	152 " "		274 seam/fab.		266 seam	
	140 " "		372 fabric		262 seam	
	-		393		265 seam	
Seam Construction						
	2" N-2582 Bias		3" A-330 Bias		3" A-330 Bias	
	A-349		A-350		A-351	
	1" N-2582 Bias		1½" A-330 Bias		1½" N-2582 Bias	
11) Cylinder Elong- ation Percent after 72 hrs.	Length Warp 1.72 Fill 1.39	Circ. 1.79 2.25	Length Warp 1.54 Fill 1.53	Circ. 2.91 2.24	Length Warp 1.85 Fill 1.87	Circ. 2.10 1.75
Inflation Pressures	11.7 psi		15.0 psi		12.5 psi	

A comparison of the results on this series of tests shows practically the same values for the used and unused material. In some instances the used material exhibited even higher breaking strength than the unused material. This may be explained by the fact that the samples had to be picked from existing airplane sections and were not all from the same fabrication run.

A slight degradation in strength can be seen after rotoflexing and creasing, i.e., approximately 2%-5% for cockpit and empennage material and about 7% for the wing material.

The permeability of hydrogen through the material increases after application of a cyclic load of 10,000 cycles by a factor of 3 to 5 for cockpit and empennage and a factor of 10 for wing material, but does not increase notably thereafter up to 100,000 cycles.

#### D. CONCLUSIONS

Airmat surface materials withstand mechanical abuse without experiencing serious loss of physical properties. After rotoflexing and creasing a strength reduction of only 2 to 5 percent for the cockpit and empennage material and approximately 7 percent for the wing fabric substantiates this fact. Additionally, it must be remembered that the tested materials were about four years old and both the empennage and wing specimen were from used panels.

The initial breaking strengths (Item 1, Table V-2) also compare favorably, considering the condition of the specimen and the method used for obtaining these values. Direct comparison of the initial strength values with those given in Appendix A cannot be used to evaluate aging effects since different

test methods were used. However, the initial strengths found in this study are sufficiently high as to preclude excessive strength losses due to aging.

On this basis then, it must be concluded that mechanical abuse does not significantly reduce the service life of the GA 468 Inflatorplane.

SECTION VI - DEAD LOAD TESTS ON WING MATERIALS

## A. GENERAL

In a further attempt to determine what effect aging had played on the Inflatoplane life, dead load time tests were performed on both the surface fabric and drop threads of the Airmat wing material. Both new and used samples were tested for comparative purposes.

From previous test results of fabric materials it is a well established fact that these materials exhibit a straight line failure curve when applied dead load is plotted as a function of time ( $\log_{10}$  scale). Furthermore, past experience has shown that although aging may shift the position of the dead load versus time curve, the slope of these curves for all practical purposes should remain constant. A large reduction in slope of this curve would imply that aging had resulted in a significant reduction of the structural capabilities in the design portion of the curve.

By definition, the quick break strength of a material is that load which when applied at a constant load rate is just sufficient to break the fabric. Any lesser load when applied to the same specimen will require a longer time to fail and will vary as a straight line function of  $\log_{10}$  time. Normal fabric design procedures select a sufficiently low load-level as to provide a long life expectancy. As an example of this, the wing of the Inflatoplane is designed to provide a life expectancy of several decades based on the design pressure, the 2.5g limit load, and the wing fabric load-time curve. This of course is an oversimplified analysis eliminating such significant

variables as aging, packaging and other material abuses which tend to reduce the life. Hence, the purpose of these tests was to determine the extent of reduction in life caused by material aging.

In order to conduct a dead load time test within a reasonable time span the fabric is loaded to some high percentage of the quick break strength (usually 50 percent or more). Such was the case for both the surface fabric and drop thread tests conducted during this study. To determine control values, quick break tests were performed prior to conducting the dead load time tests. By performing these tests first, the material quick break strength was determined and dead load was calculated based on percentages of the quick break value.

## B. SURFACE FABRIC TESTS

### (1) Control Tests

Quick-break tensile tests were performed with the Instron machine on four warp and four fill samples of surface material from new Airmat wing fabric. Samples were two inches wide, and were tested at a load rate of 12 in/min with a three-inch gage length. The average of the warp and fill strengths were used as the 100% quick-break values for dead load tests performed on additional new and used samples of the same material. The results of the control tests are given in Table IV-1.

TABLE VI-1

## Surface Fabric Quick Break Strength

<u>Direction</u>	<u>Load (Lbs/2 In)</u>	<u>Ave. Strength (Lbs/In)</u>
Warp	410,416,430,433	211
Fill	280,279,254,265	135

## (2) Dead Load-Time Tests

## (a) Apparatus and Procedure

In conducting these tests, material specimens one inch in width were dead loaded to various percentages of the quick-break strength of the new material while the time to failure was recorded. A photograph of the test setup is shown in Figure VI-1.

Some difficulty was experienced with jaw and jaw pin failures; that is, failure of the fabric at the intersection of the jaw, and inside the jaw at the clamping pin. This often required sanding or filing of the jaws as well as the intermittent use of fabric shims. The compressibility of the wooden jaws was the cause of this problem. Another complication of the results was the fact that a number of samples (used warp and all fill) were not cut parallel to the yarns. Typical of tests of this nature, a wide variation of times-to-fail was experienced and some values had to be excluded from the evaluation.

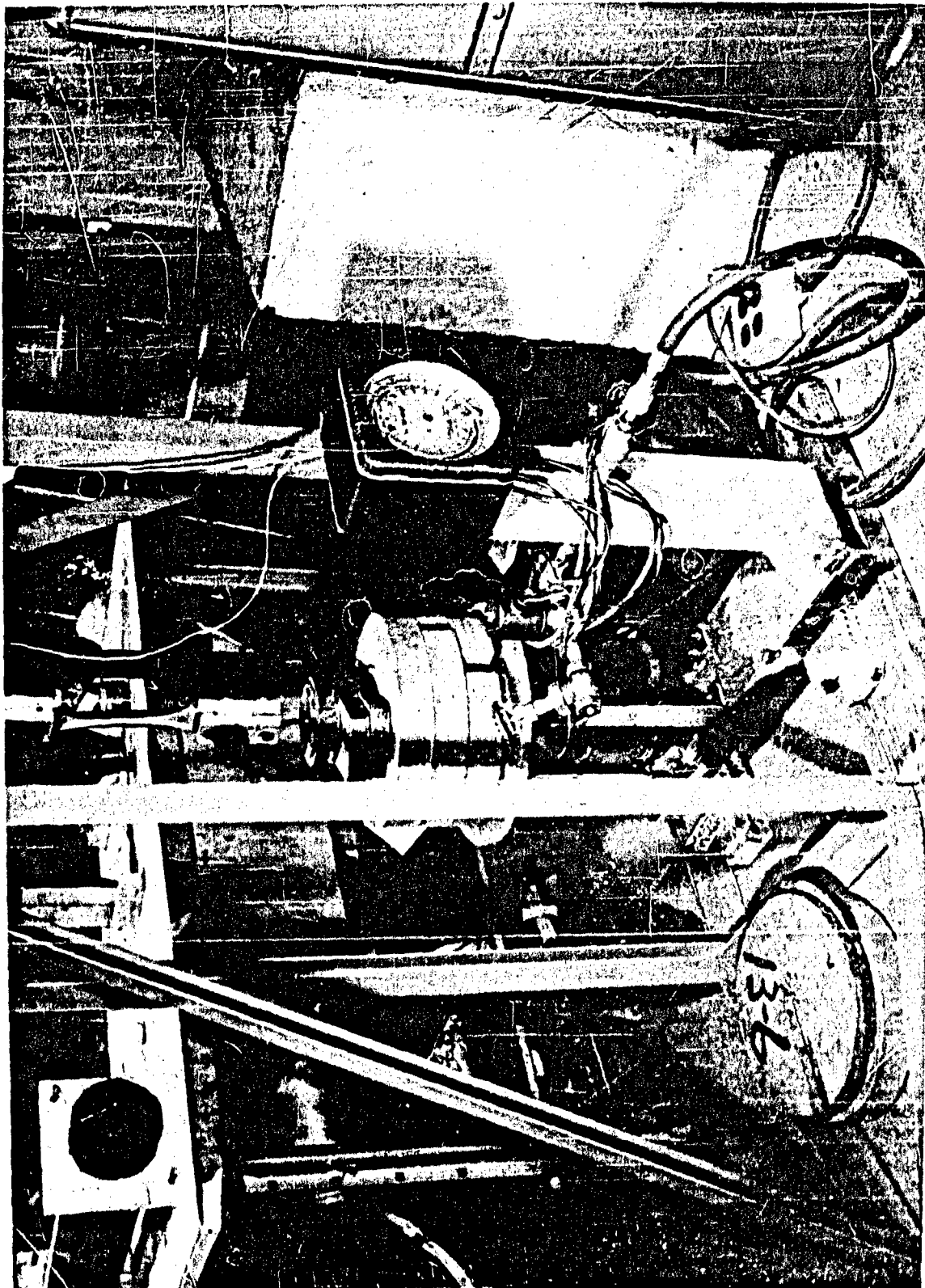


Figure VI-1 - Surface Fabric Dead Load Test Arrangement

(b) Results and Discussions

Dead load tests were conducted on samples of new and used wing surface cover fabric in both the warp and fill directions. Table V-2 presents the results of these tests. Averages are based on  $\log_{10}$  time. A plot of the results is presented in Figure VI-2.

(c) Conclusions

The time-load curves show the difference in initial strength between the new and used materials. Although some shift is evident in the position of the curves which can be attributed to the sample construction (see item a above), the fact that the slope of the curves for the corresponding new and used fabric is almost the same indicates that there is practically no change in endurance regarding life expectancy of the Inflatoplane wing surface materials.

C. DROP THREAD TESTS

(1) Control Tests

Initially, quick-break tensile tests were run on new nylon drop thread specimens using the Instron testing equipment of the Research Laboratory of the Goodyear Tire and Rubber Company. The time to break was approximately four (4) seconds at an elongation rate of 400 percent per minute with a three (3) inch gage length. The results of the tests are given in Table VI-3.

TABLE VI-2

Surface Fabric Dead Load - Time Tests

Material	Load Level %	Load (lbs/in)	Failure Time		Type of Break
New Warp	80	168 3/4		27.5 sec	Jaw*
				37.5 sec	Jaw*
			1 min	5.5 sec	Pin*
			2 min	2.5 sec	Good
			2 min	51.5 sec	Pin
			4 min	38 sec	Pin
		Average	3 min	0 sec	
	70	147 3/4	2 min	41 sec	Jaw*
			19 min	23 sec	Good
			23 min	50 sec	Jaw
			43 min	27 sec	Pin
		Average	27 min	10 sec	
New Fill	80	108		2 sec	Jaw*
				9.5 sec	Good
	75	101 1/4		14 sec	Good
			3 min	14 sec	Jaw
			8 min	44 sec	Good
			Average	1 min	52 1/2 sec
	70	94 1/2	1-3 hr (exact time unknown)		Good
	65	87 3/4	1 hr	57 1/2 sec	Jaw*
			1 hr	39 sec	Jaw*
			6 hr		Good
			15 hr		
			Average	10 hr	5 min

\*Not included in average time-to-fail

TABLE VI-2 (cont'd)

<u>Material</u>	<u>Load Level</u> <u>%</u>	<u>Load</u> <u>(lbs/in)</u>	<u>Failure</u> <u>Time</u>	<u>Type of</u> <u>Break</u>			
Used Warp	70	147 3/4		0 sec	Good		
				11.5 sec	Good		
	60	126½		23	sec	Good	
			5 min	3	sec	Good	
			5 min	47	sec	Good	
			17 min	11	sec	Good	
			Average		3 min	43½	sec
	55	116	2 min	53	sec	Good	
			7 min	50	sec	Jaw	
			1 hr	23 min		Good	
			Average		12 min	20	sec
	50	105½	21 da 21 hr	15 min		Good	
	Used Fill	75	101¼	18 min	48½	sec	Good
26 min				2½	sec	Jaw	
58 min				38	sec	Good	
1 hr				51 min	3½	sec	Good
Average				42 min	15½	sec	
65		87 3/4	11 hr	5 min		Good	
			1 da 12 hr	2 min		Jaw	
			4 da 6 hr	59 min		Jaw	
			18 da 18 hr	49 min		Good	
			Average		2 da 17 hr	37 min	

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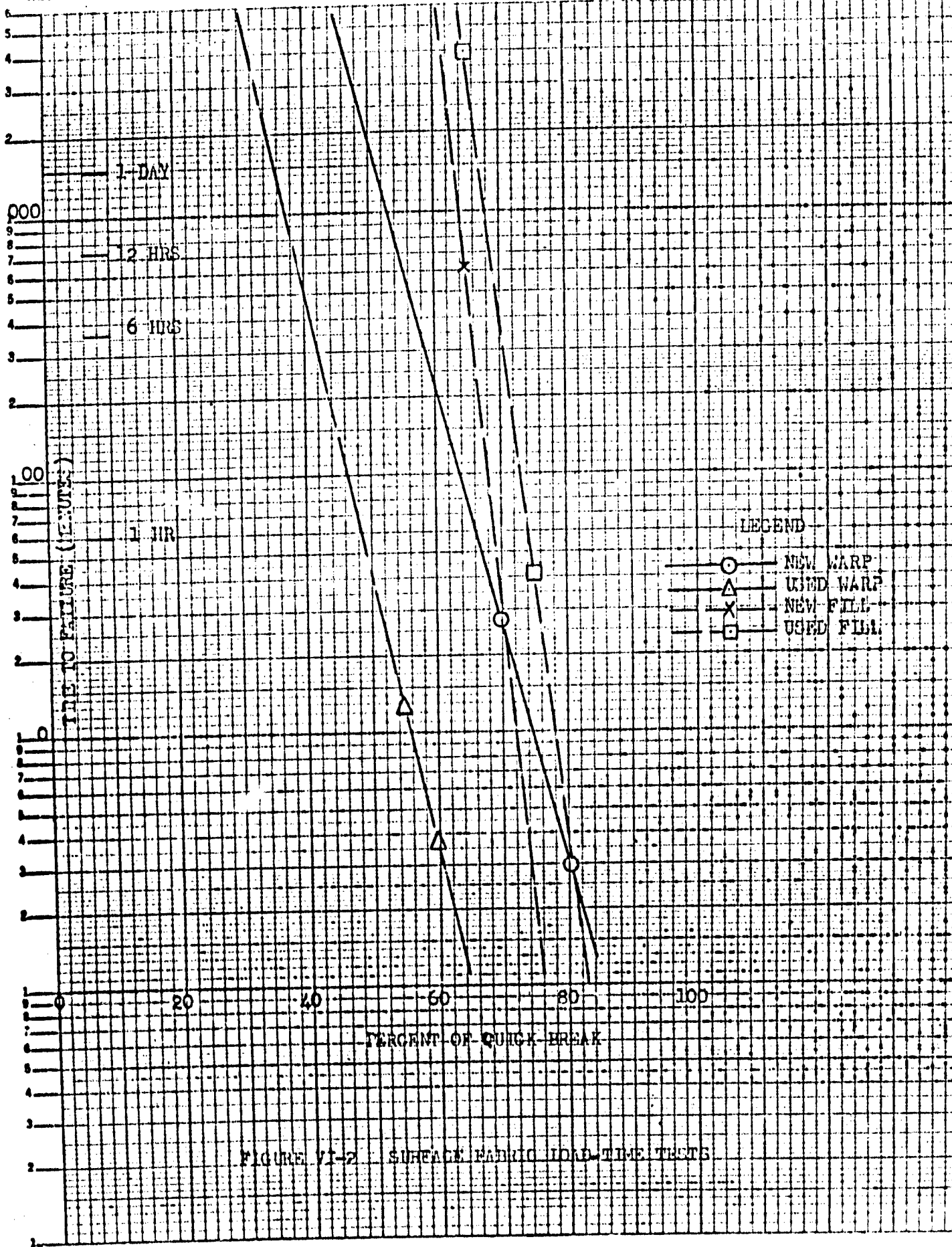


FIGURE VI-2 SURFACE FABRIC LOAD-TIME TESTS

The so-called quick-break strengths are used here merely to establish a test load level. The threads were composed of two strands of 70 denier each, twisted together. The results of Table VI-3 substantiate that the tenacity of these threads was just slightly greater than 7 gm/denier.

TABLE VI-3

Drop Thread Quick Break Strength

Specimens No.	lb
1	2.07
2	2.16
3	2.19
4	2.27
5	2.20
6	2.14
7	2.28
8	2.18
9	2.11
Average	2.18 (= 989 gm)

140 Denier = 7.06 gm/denier

(2) Drop Load-Time Tests

(a) Apparatus and Procedure

In order to perform load-time tests on the drop threads, a small wooden frame was built and steel pegs approximately 1/8 inches in diameter were fitted into the top of the frame as the upper supports for the threads. Plastic-top bottles were partly filled with mercury and were fitted with eyelets which acted as the bottom tensioning member. The diameter of the eyelets on the lower tensioner was 3/32 inches. Figure VI-3 illustrates the test apparatus.

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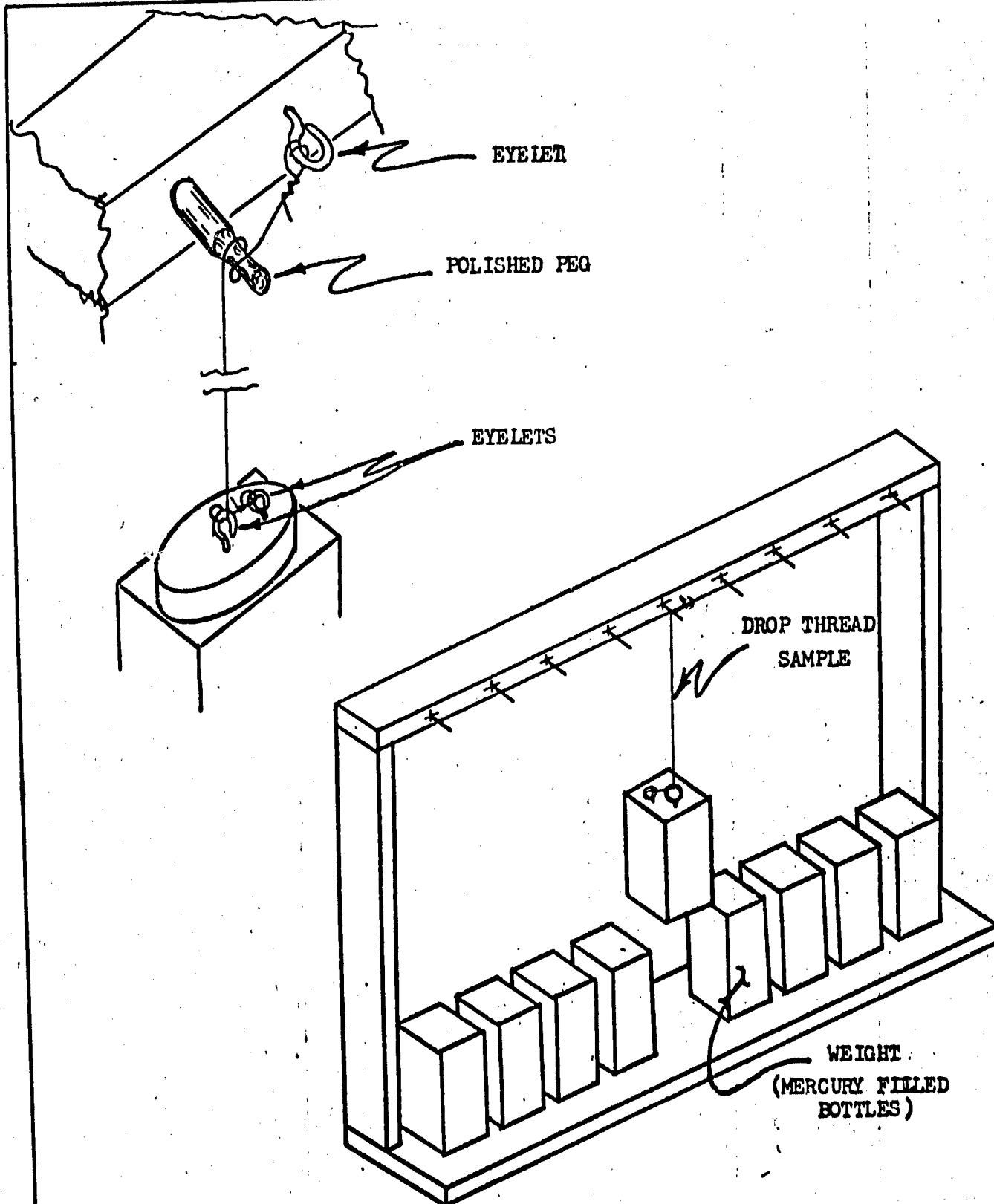


FIGURE VI-3 DROP THREAD DEAD LOAD TEST ARRANGEMENT

The threads were wound once around the top and bottom members and then tied to the small metal eyelet. The upper support pegs were sandpapered and polished with crocus cloth after the first set of tests (at 80% quick break with unused threads) was run. These involved the longest time under load. It is possible that longer times would have been obtained if these tests had also been run with polished peg supports.

The two sets of drop threads compared in dead load tests were (1) from new Code A350 Inflatoplane wing material which was recently woven to replace the GA-468 wings with "hard wrinkles" and (2) from plane No. 4111 which had seen seven hours of flight in 1959 and had recently been given a 336-hour inverted 2.5 g loading test. The second set of threads was taken from under the edge of a cable-attachment patch where the drop thread loading during the test would have been close to a maximum. The tests were run at room temperature, approximately 80° F, and the material was high-tenacity Type 66 nylon.

(b) Results and Discussion

The results of the dead load tests on new and unused threads are given in Table VI-4 and plotted in Figure VI-4.

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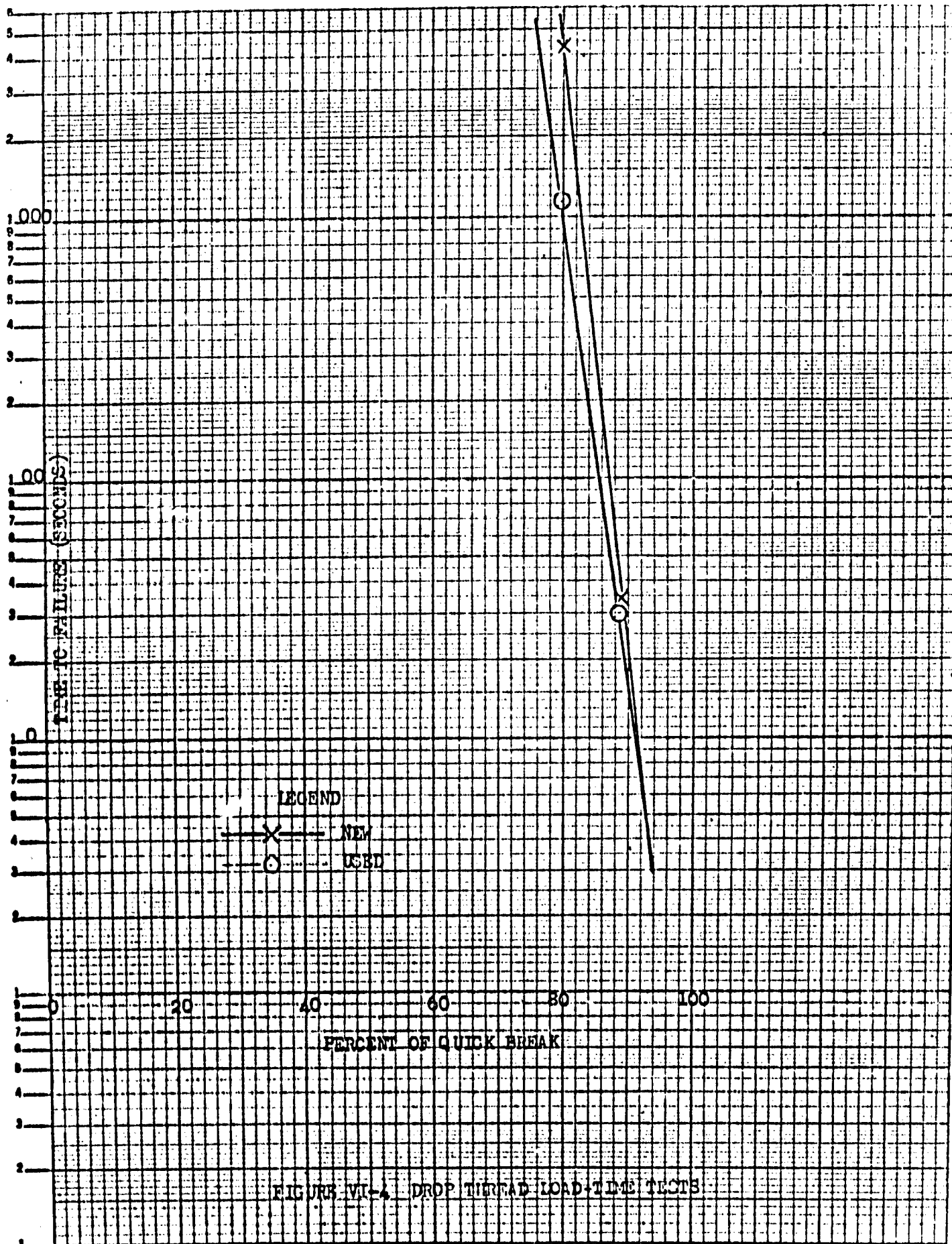


FIGURE VI-4 DROP THREAD LOAD-TIME TESTS

TABLE VI-4

Drop Thread Dead Load-Time Tests

New Wing Drop Threads

<u>Approx. % QB</u>	<u>Load gms</u>	<u>GM per Denier</u>	<u>Time to Break, Sec.</u>	<u><math>\log_{10} t_B</math></u>	<u>Ave. Time to Break</u>
80	792	5.66	3003	3.4776	
80	792	5.66	12287	4.0896	
80	792	5.66	2316	3.3647	
			Average	3.3647	4405* sec (73 m 25 sec)
89	882	6.30	15.6	1.1931	
89	882	6.30	87.7	1.9430	
89	882	6.30	28.3	1.4518	
			Average	1.5293	33.8* sec

Used Wing Drop Thread

80	792	5.66	1038	3.0160	
80	792	5.66	859	2.9338	
80	792	5.66	1699	3.2301	
			Average	3.0600	1148* sec (19 m 8 sec)
89	882	6.30	11.6	1.0645	
89	882	6.30	32.2	1.5079	
89	882	6.30	72.9	1.8627	
			Average	1.4784	30.1* sec

\*From average  $\log_{10} t_B$

As can be seen from the plot of Figure VI-4 the used threads exhibit a slight reduction in slope when percent of quick break is plotted as a function of time to failure ( $\log_{10}$ ). However, this change of slope is so slight that for all practical purposes the load carrying capability of the new and used Airmat wing drop threads is similar.

(c) Conclusions

The life of the Inflatoplane is not seriously affected by aging of the drop threads of the Airmat wing structure. From the plot of dead load versus time for new and used drop threads, the reduction in slope of the used threads is so slight that aircraft life is not seriously penalized, despite the fact that the used threads tested were in excess of 4 years old and had been subjected to considerable abuse prior to initiation of these tests.

**REFERENCES**

- II-1 GAC letter X52-764 to ONR, dated 11 November 1960.
- III-1 GER 10118 "Utility Handbook Flight Operation, Maintenance and Inspection, Inflaoplane Model GA 468", 6 March 1961.
- IV-1 MIL-A-8866 (ASG) "Aircraft Strength and Rigidity Reliability Requirements, Repeated Loads and Fatigue", dated 18 May 1960.
- IV-2 GER 9861, S97-3; "Stress Analysis of Inflatoplane Model GA 468", 18 January 1961.
- IV-3 GER 9066, "Demonstration Progress and Instrumention Report on the GA 468 Inflatoplane", 1 November 1958.
- V-1 MIL-C-21189 (AER), Amendment 1, dated 15 July 1959.